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FORMATION FLYING CONCEPT FOR CLOSE REMOTE SENSING SATELLITES

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Synthetic aperture radar (SAR) interferometry is a well-established technique based on the stereoscopic effect induced by matching SAR images obtained from slightly different orbital positions. The image resolution of present SAR interferometers may be improved by means of spacecrafts flying in close formation. In the framework of the TanDEM-X Phase-A Study, this paper discusses a suitable formation flying concept able to realize the demanding baselines for SAR interferometry, while minimizing the collision hazard associated with proximity operations. This study introduces the method, presents and validates an appropriate orbit control strategy and proposes an effective implementation of the distributed satellite concept.

INTRODUCTION

Nowadays synthetic aperture radar (SAR) interferometry is a key technology for many scientific and commercial applications aiming to perform distributed observations and high resolution imaging of the Earth. This technique is based on the stereoscopic effect that is induced by matching two SAR images obtained from two slightly different orbital positions. Whereas a differencing of SAR images obtained from two antennas separated in cross-track direction basically yields measurements of terrain elevations and therefore permits the derivation of digital elevation models (DEM), an adequate along-track separation provides measurements of the velocity of on-ground objects (e.g. for traffic monitoring, ocean currents and glacier monitoring). The accuracy of present spaceborne SAR interferometers is severely limited by either temporal de-correlation associated with repeat pass interferometry (e.g. Envisat, ERS) or by the physical dimensions of the spacecraft bus that constraints the achievable baseline length (e.g. X-SAR/SRTM Shuttle Topography mission).

These limitations may be overcome by means of two spacecrafts flying in close formation building a distributed array of sensors, where the two antennas are located on different platforms. Even though ambitious formation flying missions in low Earth orbit (LEO) have been studied throughout the past decade, the practical experience is still limited to short term proximity operations conducted in the context of the

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manned space program. GRACE, the closest LEO formation in orbit right now, operates at a separation of 200±50km at 450km altitude[1]. The EO-1/Landsat formation flying demonstration was carried out at 700km altitude with a separation of 450±85km over a 5 months time span[2]. Both missions require only infrequent orbit adjustment maneuvers and allow for a convenient ground control based on a differential drag mechanism.

With this background, the TanDEM-X/TerraSAR-X (TDX/TSX) mission has recently been proposed in a contest for new Earth observation missions within the German national space program[3]. It involves two almost identical SAR-satellites, with a size of 5x2.4m, a mass of 1200kg, carrying a high-resolution SAR operating in the X-band (9.65GHz). The two spacecrafts will fly in a precisely controlled formation to form a radar interferometer with typical baselines of 1km, and will be operated for a period of 5 years in a nearly constant 514km sun-synchronous dusk-dawn orbit with 97° inclination and an 11 day repeat period. In the framework of a related Phase-A study, this paper discusses a suitable formation flying concept able to realize the demanding baseline for SAR interferometry, while minimizing the collision hazard associated with proximity operations.

The specific orbital configuration to be chosen for the TSX/TDX formation is driven by the need for achieving a certain effective baseline (<4km), defined as the projection of the distance between the two satellites onto the plane spanned by the flight and target directions, within a maximum along-track separation (<2km), and for ensuring safe proximity operations. It will be shown that both requirements can jointly be met by the concept of eccentricity-/inclination- (e-/i-) vector separation. This technique has originally been developed for the safe collocation of geostationary (GEO) satellites and is here extended to LEO formations. In the presence of along-track position uncertainties, a collision hazard can be avoided by proper separation of the two spacecrafts in radial and cross-track directions. This implies a coordinated selection of the relative orbital elements which results in an elliptic relative motion perpendicular to the flight direction. The paper briefly describes the linearized mathematical formulation of the adopted relative motion model. In particular a description in terms of Keplerian element differences is preferred to a Cartesian formulation for proximity analyses and orbit control purposes.

Once nominal orbital parameters for the formation have been established, it is necessary to study the stability of the relative motion in order to investigate a relative orbit control strategy. The relevant orbital perturbations are considered in this paper, and the effects on the relative motion are shown. While periodic perturbations of the orbital elements cancel out for satellites operating in close proximity, secular perturbations of the relative e-/i- vectors tend to disturb an initial nominal configuration. This paper presents a convenient orbit control concept to maintain the safe e-/i- vector separation of the TSX/TDX formation at an affordable expenditure in terms of thruster activations and propellant consumption. The problem has two important features. First of all, thruster activities have to be minimized in order to maximize the available time for SAR data collection, which renders a continuous control of the relative orbit inadequate. In addition, the leader satellite (TSX) carries out an independent repeat pass interferometry mission and is therefore completely passive from a formation control point of view. Routine orbit maintenance maneuvers are performed on TSX to ensure that the actual spacecraft orbit deviates by less than 250m (perpendicular to the flight direction) from a predefined Earth-fixed reference trajectory. As a result, on top of duplicating TSX maneuvers, the chaser satellite (TDX) will be required to perform corrective maneuvers for maintaining the nominal e-/i- vector separation.

TDX/TSX - AN X-BAND SAR FORMATION

TSX is an advanced SAR-satellite system for scientific and commercial applications, which is realized in a public-private partnership between DLR and Astrium GmbH. A Russian DNEPR-1 rocket launched from Baikonour, Kazakhstan, will inject TSX into a 514 km sun-synchronous dusk-dawn orbit with 97° inclination and an 11 day repeat period. The launch is presently scheduled for May 2006. TSX is planned to be operated for a period of 5 years and will therefore provide SAR-data on a long-term, operational basis. The German Space Operation Center (GSOC) will provide the Mission Operations Segment (MOS) using ground stations at Weilheim and Neustrelitz.

The TSX mission will provide the scientific community with high-quality, multi-mode X-band SAR-data for scientific research and applications. Furthermore, it will support the establishment of a commercial EO-market and help to develop a sustainable Earth observation (EO) service business in Europe, based on TSX derived information products. The broad spectrum of scientific application areas includes hydrology, geology, climatology, oceanography, environmental monitoring and disaster monitoring as well as cartography with interferometry.

As a complement to TSX, the TDX mission has recently been proposed in a contest for new Earth observation missions within the German national space program. It involves a second spacecraft, which is almost identical to TSX and shall likewise be operated for five years. The two spacecraft will fly in a precisely controlled formation to form a radar interferometer with typical baselines of 1 km. This allows a much higher resolution than achievable in the X-SAR/SRTM Shuttle Topography mission and thus the generation of digital elevation models (DEMs) with unrivaled accuracy.

Formation Flying Requirements

From a misson perspective, the following top level requirements for the close formation flying of TSX and TDX satellites can be formulated[4]:

- During interferometric imaging the two spacecraft shall be separated by an effective baseline of 500-4000m (measured in the direction perpendicular to the plane spanned by the flight direction and the target direction).
- The along-track separation during data takes shall be less than 2000 m. Ideally, the along-track separation should vanish.
- Proximity operations of TSX and TDX shall be conducted such as to minimize the risk of a collision.

- The TDX mission concept shall have minimum impact on the existing design of the TSX satellite.

In the sequel, these requirements are used to establish a preliminary formation flying concept.

FLIGHT DYNAMICS OF CLOSE FORMATIONS

In order to predict the relative motion of two objects in space, a rigorous approach is to numerically integrate the equations of motion in the presence of all relevant perturbations. The relative motion may then be derived from a difference of the absolute trajectories and a suitable mapping into the co-moving orbital reference frame. In contrast to an elaborate numerical integration of the orbit followed by a subsequent differencing of individual trajectories, the equations of motion for the two-body problem may directly be differenced. In the limiting case of circular orbits and close formations, a closed-form of relative motion equations may be derived known as the Hill-Clohessy-Wiltshire equations. Despite the analytical solution the Cartesian formulation does not provide immediate insight into some aspects of the relative motion. A description in terms of Keplerian element differences is therefore preferable for proximity analyses. To this end, the concept of eccentricity and inclination vectors is introduced and a formulation suitable for LEO satellites is adopted. The relative motion is then described in terms of the relative e/i-vectors[5].

Linear Relative Motion Model

For two formation flying spacecraft (k=1,2) with orbital elements semi-major axis a_k, eccentricity e_k, inclination i_k, right ascension of ascending node Ω_k, argument of perigee ω_k and mean anomaly M_k, we may define the relative eccentricity and inclination vector via the relations

$$\Delta\boldsymbol{e} = e_2\begin{pmatrix}\cos\omega_2\\ \sin\omega_2\end{pmatrix} - e_1\begin{pmatrix}\cos\omega_1\\ \sin\omega_1\end{pmatrix} \tag{1}$$

and

$$\Delta\boldsymbol{i} = \begin{pmatrix} i_2 - i_1 \\ (\Omega_2 - \Omega_1)\sin i_1 \end{pmatrix} = \begin{pmatrix}\Delta i\\ \Delta\Omega\sin i_1\end{pmatrix}. \tag{2}$$

Both spacecraft are assumed to fly in a LEO near-circular orbit sufficiently close to each other to justify a linearization of the relative motion equations. Depending on the specific application, either a Cartesian or a polar representation of the relative e-/i-vectors is preferable, for which we employ the following notation:

$$\Delta\boldsymbol{e} = \begin{pmatrix}\Delta e_X\\ \Delta e_Y\end{pmatrix} = \delta e\begin{pmatrix}\cos\varphi\\ \sin\varphi\end{pmatrix} \tag{3}$$

and

$$\Delta\boldsymbol{i} = \begin{pmatrix} \Delta i_X \\ \Delta i_Y \end{pmatrix} = \delta i \begin{pmatrix} \cos\theta \\ \sin\theta \end{pmatrix}. \tag{4}$$

With otherwise identical orbital elements a pure eccentricity vector separation results in a relative orbit of s/c-2 with respect to s/c-1 that is an ellipse of semi-minor axis $a\delta e$ in radial direction (i.e. $\boldsymbol{e}_R$, positive outwards) and semi-major axis $2a\delta e$ in along-track direction (i.e. $\boldsymbol{e}_T$, positive in the direction of the satellite motion). Likewise, a pure inclination vector separation results in harmonic oscillation of magnitude $a\delta i$ in cross-track direction (i.e. $\boldsymbol{e}_N$, perpendicular to the orbital plane). While δe and δi determine the amplitude of the relative in-plane and out-of-plane motion, its instantaneous phase is determined by the polar angles φ and θ of the relative e-/i-vectors (Figure 1). The angle φ indicates the relative perigee: whenever the mean argument of latitude (i.e. $u = \omega + M$) of s/c-2 equals φ the spacecraft are at their maximum radial separation. The angle θ instead indicates the argument of latitude at which s/c-2 crosses the orbital plane of s/c-1 in ascending direction (i.e. the relative ascending node). Aside from the eccentricity and inclination vector difference, the relative motion of two spacecraft is affected by differences $\Delta a = a_2 - a_1$ and $\Delta u = u_2 - u_1$ in semi-major axis and mean argument of latitude. These result in a systematic offset of size Δa in radial direction as well as a drift of $-3\pi\Delta a$ per revolution and a constant offset $a_1\Delta u$ in along-track direction.

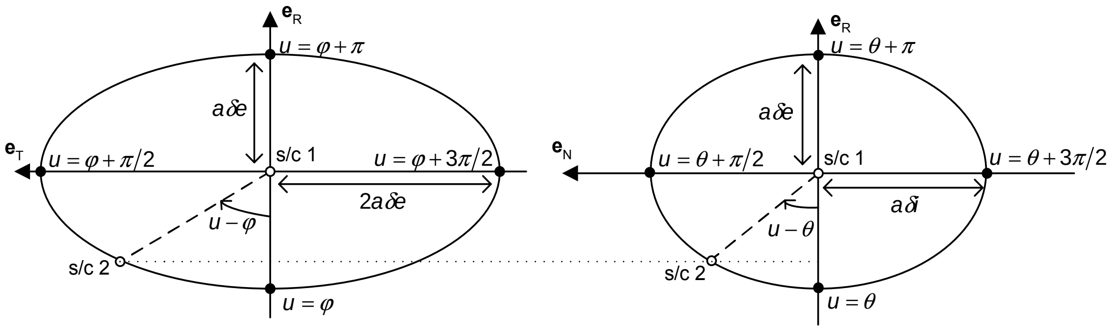

Figure 1. Relative motion of two spacecraft induced by $\Delta\boldsymbol{e}$ and $\Delta\boldsymbol{i}$ vectors.

Overall, the relative position vector $\Delta\boldsymbol{r}$ of s/c-2 with respect to s/c-1 in a local horizontal frame aligned with the radial ($\boldsymbol{e}_R$), along-track ($\boldsymbol{e}_T$) and cross-track ($\boldsymbol{e}_N$) directions, can thus be described by the linearized equation[5]

$$\begin{pmatrix} \Delta r_R / a \\ \Delta r_T / a \\ \Delta r_N / a \end{pmatrix} = \begin{bmatrix} \Delta a / a & 0 & -\Delta e_X & -\Delta e_Y \\ \Delta u & -3\Delta a / 2a & -2\Delta e_Y & +2\Delta e_X \\ 0 & 0 & -\Delta i_Y & +\Delta i_X \end{bmatrix} \begin{pmatrix} 1 \\ u - u_0 \\ \cos u \\ \sin u \end{pmatrix}, \tag{5}$$

where $u \equiv u_2$ is the independent variable, u_0 is the mean argument of latitude at the epoch of the orbital elements (i.e. the initial epoch t_0), $a \equiv a_1$ is the semi-major axis of s/c-1. Differentiation with respect to time furthermore yields the relative velocity vector $\Delta\boldsymbol{v}$ of s/c-2 with respect to s/c-1 as[5]

$$\begin{pmatrix} \Delta v_R / v \\ \Delta v_T / v \\ \Delta v_N / v \end{pmatrix} = \begin{bmatrix} 0 & 0 & -\Delta e_Y & +\Delta e_X \\ -3\Delta a / 2a & 0 & +2\Delta e_X & +2\Delta e_Y \\ 0 & 0 & +\Delta i_X & +\Delta i_Y \end{bmatrix} \begin{pmatrix} 1 \\ u - u_0 \\ \cos u \\ \sin u \end{pmatrix}, \tag{6}$$

where v denotes the orbital velocity in a circular orbit of radius a.

Earth Oblateness Perturbations

The asphericity of the Earth results in a variety of short periodic, long-periodic and secular perturbations of the orbital elements of a LEO satellite[6]. For formation flying satellites operating in close proximity, the periodic perturbations are essentially cancelled, leaving a secular change of the relative e- and i-vectors. Ignoring short periodic perturbations, the relative eccentricity vector

$$\Delta\boldsymbol{e} = \delta e \begin{pmatrix} \cos(\varphi_0 - \dot{\varphi}\, t) \\ \sin(\varphi_0 - \dot{\varphi}\, t) \end{pmatrix} \tag{7}$$

evolves along a circle of radius δe that is centered in the origin of the e-vector plane and traversed at an angular velocity of $\dot{\varphi}$. The associated period

$$T_G = \frac{2\pi}{\dot{\varphi}} \approx \frac{4}{3} T \frac{a^2}{R_\oplus^2} \frac{1}{J_2 \left| 5\cos^2 i - 1 \right|} \tag{8}$$

is determined by the second order zonal coefficient J_2 and the equatorial radius $R_\oplus$ of the Earth. For TSX and TDX with an orbital altitude of 514 km, an inclination i = 97.42° and an orbital period $T \approx 95$ min, eq. (8) yields a period T_G pf roughly hundred days. The relative inclination vector

$$\Delta\boldsymbol{i} = \begin{pmatrix} \Delta i \\ \Delta\Omega \sin i - \frac{3\pi}{T} J_2 \frac{R_\oplus^2}{a^2} \sin^2 i \cdot \Delta i \end{pmatrix}, \tag{9}$$

is likewise affected by J_2-perturbations that cause a secular shift of the orbital planes and thus a linear drift of the Δi_y component. The drift rate depends on the inclination difference. The secular motion of the relative e-/i-vectors caused by the Earth oblateness would ultimately destroy an initial relative configuration unless correction maneuvers are performed.

Differential Drag Perturbations

Ignoring density variations over distances of less than a few kilometers, the differential acceleration due to atmospheric drag can be modeled as

$$\Delta a_D = \frac{1}{2}\left(C_{D1}\frac{A_1}{m_1} - C_{D2}\frac{A_2}{m_2} \right)\rho v_a^2 , \tag{10}$$

where ρ denotes the atmospheric density, v_a the spacecraft velocity with respect to the atmosphere, $B_k = C_{Dk}A_k/m_k$ the ballistic coefficients (i.e. the effective area-to-mass ratio) of the spacecraft (k=1,2). Based on the experience gained in the GRACE (Gravity Recovery and Climate Experiment)[1] mission, it may be assumed that the nominal ballistic coefficients of TSX and TDX with their nearly identical design can be matched to roughly one percent near the start of the TDX mission. Given an overall fuel mass of roughly 5% for orbit acquisition, orbit keeping and de-orbiting, mass variations during the mission lifetime may contribute an additional estimated difference of 1%. During nominal operations, the attitude of both spacecraft is aligned with the along-track direction in the Earth-fixed frame to better than 1 arcmin. Attitude control induced variations of the ballistic coefficients are therefore confined to less than 0.1% and can be neglected in the following considerations. A conservative assumption for the offset in the ballistic coefficients is therefore $\varepsilon = (B_2\text{-}B_1)/B_1$ = 2%.

According to the Harris-Priester model[7], the atmospheric density at the altitude of TSX and TDX may amounts to roughly 1 g/km^3 for mean solar flux conditions. At a ballistic coefficient B_1 = 0.006 m^2/kg and an orbital velocity v = 7.6 km/s the differential acceleration amounts to roughly 3 nm/s^2. This causes an accumulated along-track offset of 5 cm within one orbital revolution and a 10 m offset after one day. Even though these values might increase by a factor of ten during high solar activities and geomagnetic storms, differential drag has evidently little to no impact on the TSX/TDX formation control during nominal operation. On the other hand, this conclusion is no longer valid if one of the two spacecraft enters a safe mode with uncontrolled yaw angle. The effective cross-section might then increase by a factor of three and thus cause large differential drag accelerations of several hundred nm/s^2. When lasting over extended periods of time, a safe mode may thus cause a notable change in the along-track separation (up to several kilometres within one day). This gives great importance to the collision avoidance strategy. The nominal formation configuration shall tolerate either large prediction uncertainties or extreme changes of the along-track separation without collision hazard. In such a way the nominal separation can be restored by a series of corrective maneuvers performed either autonomously or with ground intervention after the end of the safe mode.

ECCENTRICITY/INCLINATION VECTOR SEPARATION

The specific orbital configuration to be chosen for the TDX/TSX formation is driven by the need for achieving a certain effective baseline and for ensuring safe proximity operations. Both requirements can jointly be met by the concept of e-/i-vector separation. It implies a co-ordinate selection of the relative orbital elements which results in an elliptic relative motion perpendicular to the flight direction. Applicable orbital parameters for the TDX/TSX formation are established in the following section.

Collision Avoidance

The concept of e/i-vector separation has originally been developed for the safe collocation of GEO satellites[6] but can likewise be applied for proximity operations in LEO formations. It is based on the consideration that the uncertainty in predicting the along-track separation of two spacecraft is generally much higher than for the radial and cross-track component. Due to the coupling between semi-major axis and orbital period, small uncertainties in the initial position and velocity result in a corresponding drift error and thus a secularly growing along-track error. Predictions of the relative motion over extended periods of time are therefore particularly sensitive to both orbit determination errors and maneuver execution errors.

To avoid a collision hazard in the presence of along-track position uncertainties, care must be taken to properly separate the two spacecraft in radial and cross-track direction. This can be achieved by a parallel (or anti-parallel) alignment of the relative e-and i-vectors. Even though these vectors are differently defined for near-equatorial, geostationary satellites and the highly inclined low Earth orbits, the convention adopted here ensures consistency with [6] and the same considerations are therefore applicable.

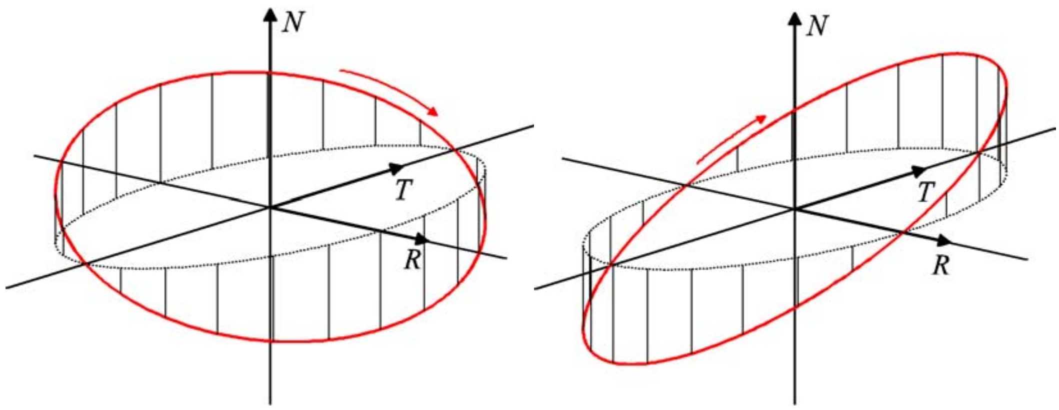

Figure 2. Relative motion for parallel (right) and orthogonal (left) relative e-/i-vectors

Using the notation introduced above, parallel $\Delta\boldsymbol{e}$ and $\Delta\boldsymbol{i}$ imply equality of the phase angles φ and θ, or in other words coincidence of relative perigee (i.e. maximum radial separation) and relative argument of latitude (i.e. zero cross-track separation). In contrast to this, for orthogonal $\Delta\boldsymbol{e}$ and $\Delta\boldsymbol{i}$ the radial and cross-track separation may jointly vanish, which is risky in case of pronounced along-track position uncertainties (Figure 2). The inter-satellite distance is always ensured to be larger than min($a\delta e$, $a\delta i$) even in the case of vanishing along-track separation. Smaller thresholds may only be encountered in case of drifting satellites, where the radial offset Δa needs to be accounted as well. In general, this can be compensated by a suitably increased eccentricity vector separation.

Nominal TDX/TSX Relative Orbit

The objective of satellites formation design is to develop fuel-efficient relative spacecraft trajectories that are useful for synthesizing scientific instruments. Due to the fact that thruster activities have to be minimized in order to maximize the available time for SAR data collection, these trajectories are thrust-free and are referred to as passive apertures. The height accuracy of interferometric images is determined by the so-called effective baseline[8]. The effective baseline is the distance between the two planes spanned by the flight direction (along-track) and the target direction (antenna-beam pointing direction) of the respective satellite. While a larger effective baseline provides better height accuracies of the desired scene, small baselines assure unambiguous retrieval of the height information with successful phase unwrapping. As a consequence, the formation flying concept has to allow interferometric data acquisition with large and small baselines at a fixed baseline ratio.

Considering both safety and imaging constraints, the TDX mission requirements can be fulfilled by a formation with parallel relative e-/i-vectors. Nominal relative orbital elements can be defined in accordance with the above considerations. The relative eccentricity modulus is limited by the minimum radial separation necessary for safe proximity operations and the maximum along-track separation required by SAR interferometry: 200m $\leq a\delta e \leq$ 800m. Similarly the relative inclination modulus is driven by the requirements on operational safety and effective baseline: 500m $\leq a\delta i$ $\leq$ 3000m. To avoid a secular motion of the relative inclination vector the absolute inclinations of both spacecraft should be identical ($i_1 = i_2$). In this case, a separation of the two orbital planes by angle δi is achieved through a small offset $\Delta\Omega$ in the right ascensions of their ascending nodes. The resulting relative inclination vector has a phase angle $\theta = \pm\pi/2$ and the same (or opposite phase angle) must be selected for the relative eccentricity vector:

$$\Delta\boldsymbol{e} = \begin{pmatrix}0 & \pm\delta e\end{pmatrix}^T \quad ; \quad \Delta\boldsymbol{i} = \begin{pmatrix}0 & \pm\delta i\end{pmatrix}^T . \tag{11}$$

The spacecraft achieve their largest cross-track separation at the equator, where the radial separation vanishes. Vice-versa the radial separation is maximum near the poles, where the two orbital planes intersect (see Figure 1). The relative semi-major axis, Δa, is nominally zero. It must be kept as small as possible to prevent an along-track drift. Finally, the relative argument of latitude, Δu, should nominally be zero and is limited by the maximum along-track separation for the formation. If necessary, a non-zero value of Δu could be selected to achieve a zero along-track separation at a specified argument of latitude.

RELATIVE ORBIT CONTROL

The maintenance of a formation within its control window requires the performance of frequent maneuvers. Among the various control methods and algorithms, we apply a simple bang-bang strategy, primarily to estimate the required amount of velocity increment for the TDX mission. Although the considered mission will require data takes only for a fraction of the orbital period, the number of conducted maneuvers should be kept at a minimum. Especially during data-takes, maneuvers must not be implemented, since un-modeled performance errors would spoil the precise relative distance measurements[9]. The use of thrusters is preferred which deliver a measurable force in a limited time span, while no thruster activity is required during most of the orbit.

Deterministic Maneuver Planning

In order to maintain the nominal orbital configuration of the TDX/TSX formation, regular maneuvers will have to be executed by the TDX spacecraft. Primarily, these maneuvers must compensate for changes of the relative orbit caused by TSX orbit keeping maneuvers and changes of the relative orbit caused by the Earth oblateness. In addition, occasional maneuvers may be required to intentionally change the relative orbit (e.g. to achieve different effective baselines for SAR interferometry) or to restore the nominal configuration after a contingency. As discussed in the previous section, differential drag is of minor relevance during normal operations and similar considerations apply for differential gravity.

Routine orbit keeping maneuvers are performed on TerraSAR-X to ensure that the actual spacecraft orbit deviates by less than 250 m (perpendicular to the flight direction) from a pre-defined Earth-fixed reference trajectory[10]. Aside from compensating semi-major axis changes due to atmospheric drag, these maneuvers maintain a near-frozen eccentricity vector for TSX. Depending on solar activity the size and frequency of TSX orbit keeping maneuvers vary between 1 cm/s about once per week near the start of mission and 5 cm/s approximately once per day near the end of life[11].

Given the fact that a 5 cm/s along-track maneuver changes the semi-major axis by 100 m and thus introduces a drift of 1000 m per revolution (or 15 km/d), it is evident that TDX is required to perform closely synchronized orbit keeping maneuvers for maintaining the TSX/TDX formation geometry. Ideally, TSX maneuvers should be duplicated by TDX, in which case the relative orbital elements of both spacecraft won't be affected.

On top of duplicating TSX maneuvers, the TDX spacecraft will be required to perform corrective maneuvers for maintaining the relative eccentricity vector. As previously explained the rate of change is always perpendicular to the relative eccentricity vector itself. Given a nominal relative perigee at $\varphi = \pi/2$, the vector $\Delta\dot{\boldsymbol{e}}$ is always directed along the positive e_x-axis. It can thus be counteracted by performing along-track maneuvers near the ascending or descending node of the orbit. Given the period $T_G \approx 100$d of the J_2 induced perigee variation, the daily shift of the relative eccentricity vector amounts to roughly $0.06 \cdot \delta e$, see eq. (7-8). Assuming a nominal eccentricity offset of $a\delta e$ = 300m, a total velocity increment of 1 cm/s will thus be required each day to maintain the nominal relative perigee. This has to be split symmetrically into positive and negative along-track velocity increments (executed at the descending and ascending node of the orbit, respectively) to avoid a net change of the semi-major axis. When performing two burns of 0.5 cm/s separated by half a revolution, a semi-major offset of 10 m is temporarily introduced and the mean along-track position will thus be shifted by roughly 50 m. This appears tolerable and could even be used for intentional orbit adjustments. Alternatively, a larger number of maneuvers (e.g. 0.3 mm/s at each nodal crossing) may be executed to minimize the impact of eccentricity control on the along-track separation of the formation.

Short-term Numerical Simulation

The sequel focuses on a numerical validation of the formation flying concept. The nonlinear simulation makes use of state-of-the-art dynamic models for a numerical integration of the equations of motions[12-13], including geo-potential forces to degree and order 40 (GGM01S GRACE gravity model), sun and moon third body perturbations, earth tides, atmospheric drag (Jacchia/Gill density model) and solar radiation pressure. The first short-term simulation covers a time interval of 25 day. The representative initial nominal configuration selected for the TDX/TSX formation flying is given by

$$\Delta a = \Delta u = 0 \quad ; \quad a\Delta\boldsymbol{e} = \begin{pmatrix}0 & +300\text{m}\end{pmatrix}^T \quad ; \quad a\Delta\boldsymbol{i} = \begin{pmatrix}0 & -600\text{m}\end{pmatrix}^T . \qquad (12)$$

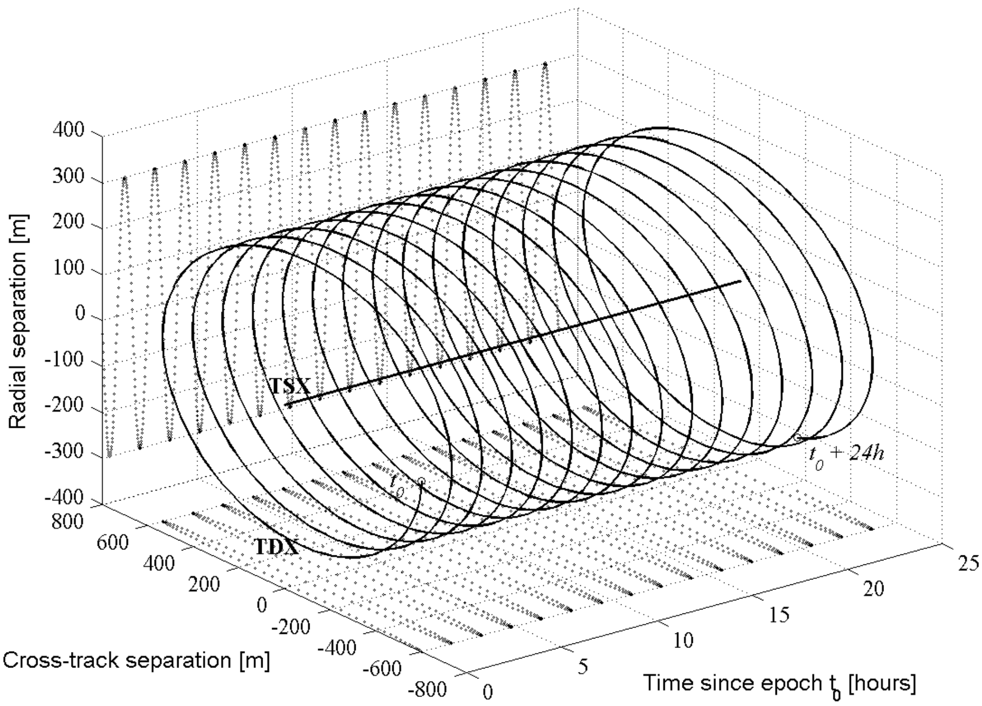

Figure 3. Osculating relative motion of TDX w.r.t. TSX in radial and cross-track directions during the first 24 hours. Nominal configuration.

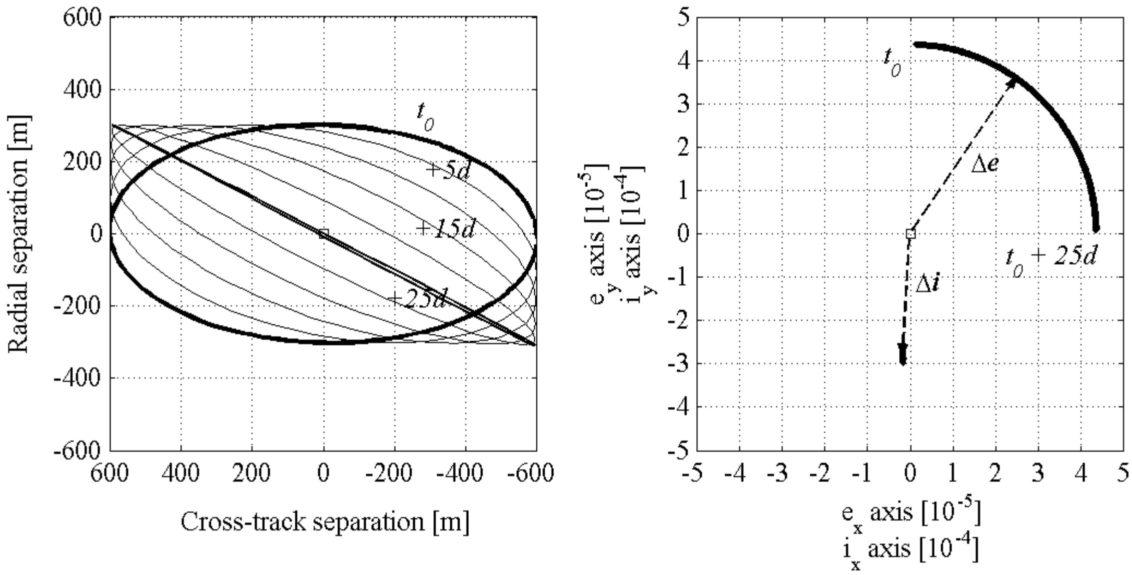

Figure 4. Osculating relative motion of TDX w.r.t. TSX in radial and cross-track directions sampled every 5 days (left). Mean relative e-/i-vectors over 25 days (right). Uncontrolled motion.

Figure 3 shows the osculating relative motion in the plane perpendicular to the flight direction (i.e. cross-track and radial directions). The formation configuration is stable over 24 hours and shows the potentiality of a proper e-/i-vector separation in synthesizing passive apertures. Unfortunately the secular J_2-perturbations tend to change the orientation of the relative eccentricity vector, whereas the relative inclination vector is nearly constant because of the relative perigee at 90°. Figure 4 shows the motion of the mean relative e-/i-vectors over 25 days and the associated evolution of the relative motion in radial and cross-track directions. As expected from eq. (8), $\Delta\boldsymbol{e}$ draws a quarter of a circle-arc in 25 days (i.e. $T_G \approx 100$d), thus the angle formed by the relative e-/i-vectors amounts to 90° at the end of the simulation. The collision hazard is significant (see Figure 4), especially in presence of along-track separation uncertainties.

Long-term Numerical Simulation

The second numerical simulation covers a period of 90 days. The objective is to apply the proposed relative orbit control strategy, and verify the required amount of velocity corrections. As soon as the angle formed by the relative e-/i-vectors reaches a predefined threshold of 7°, two along-track maneuvers, separated by half a revolution, are executed. The size and location of the velocity corrections are given by

$$\Delta v_T^1 = +\frac{v}{4}\left(\left\|\Delta\boldsymbol{e}^c\right\| + \frac{\Delta a^c}{a}\right) \quad ; \quad \Delta v_T^2 = -\frac{v}{4}\left(\left\|\Delta\boldsymbol{e}^c\right\| - \frac{\Delta a^c}{a}\right) \tag{13}$$

and

$$u^1 = \mathrm{atan}\left(\frac{\Delta e_Y^c}{\Delta e_X^c}\right) \ ; \ u^2 = u^1 + \pi \ , \tag{14}$$

where the superscript ^{c} indicates the desired correction of the relative orbital elements. Differential drag (i.e. ε = 2%) induces a pronounced drift of the along-track separation that has to be counteracted. To this purpose, a proper adjustment of Δa^c is introduced, that limits the variation of the mean argument of latitude within a control window of ±60m. Figure 5 shows the osculating controlled motion in the plane perpendicular to the flight direction (i.e. cross-track and radial directions), while Figure 6 depicts the relative motion in the other directions (i.e. along-track/cross-track and along-track/radial directions). The formation is maintained over 3 months, and the angle formed by the relative e-/i-vectors oscillates between 0° and 7° with a period of roughly 2 days (cf. $\dot{\varphi} \approx 360°/100\text{d}$). As a consequence the maneuver cycle is approximately 2 days, and the daily maneuver budget for relative orbit keeping is roughly 1 cm/s (Figure 7).

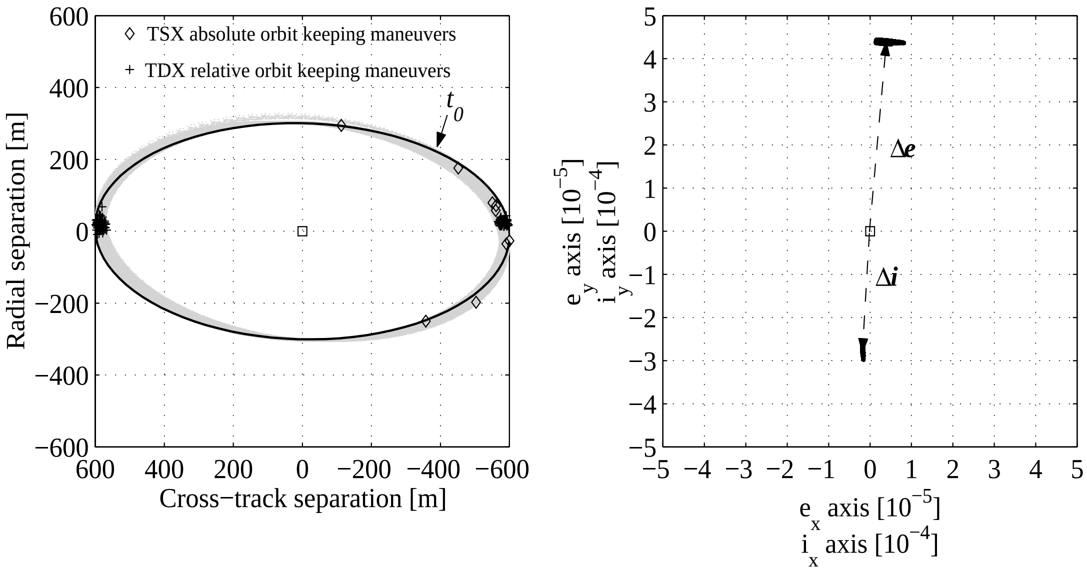

Figure 5. Osculating relative motion of TDX w.r.t. TSX in radial and cross-track directions with maneuver locations (left). Mean relative e-/i-vectors (right). Controlled motion over 90 days.

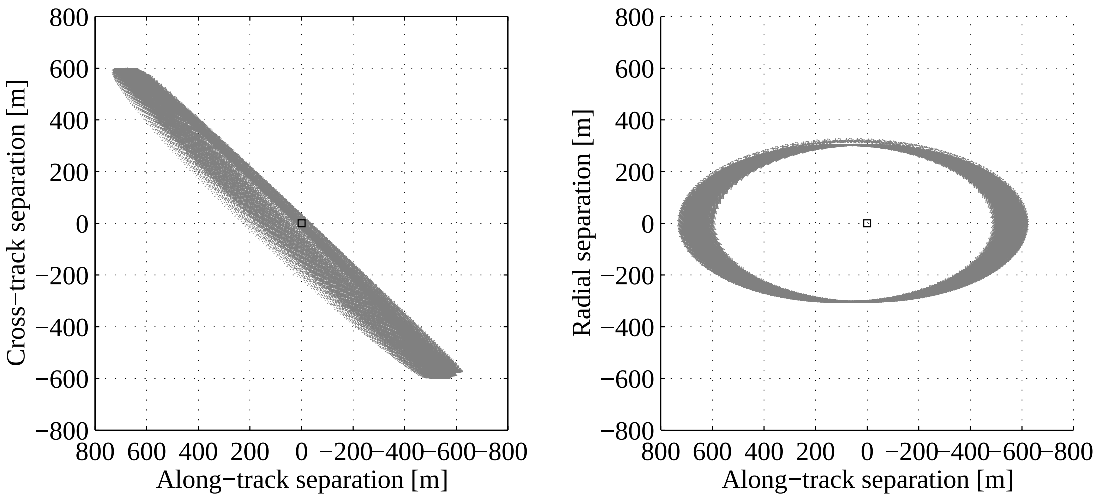

Figure 6. Osculating relative motion of TDX w.r.t. TSX in along-track/cross-track directions (left), along-track/radial directions (right). Controlled motion over 90 days.

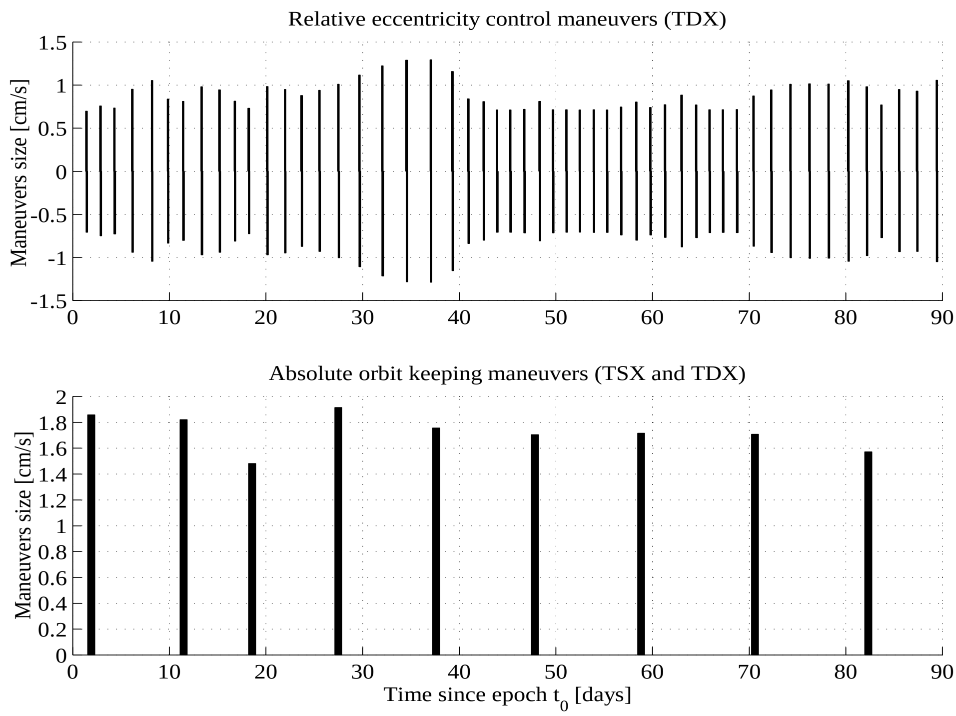

Figure 7. Tangential velocity corrections for relative orbit control (up) and TSX orbit keeping (down).

OPERATIONAL IMPLEMENTATION

The control of satellite formations is performed by an activation of appropriate onboard thrusters. The TSX spacecraft is equipped with a Hydrazine mono-propellant propulsion system. The actuators are arranged in two branches with four thrusters each and will be used for the orbit maintenance maneuvers. The nominal thrust level for each thruster is 1 N at begin of life. Apart from two major modifications, TDX will be a 1:1 rebuild of TSX. This ensures operational compatibility and guarantees minimum impact on the present TSX design. First of all an Inter-Satellite Link (ISL) must be established. TSX will continuously send House-Keeping (HK) telemetry data in S-band (low rate nominal, high rate during ground-station contacts). TDX will receive TSX telemetry by using an additional S-band receiver system. Secondly the new requirements on the relative orbit control impose an adaptation of the propulsion system. Instead of modifying the present TSX Hydrazine System to cope with an increased propellant consumption, with a larger number of cycles and smaller thrust levels, an improved solution is to equip TDX with a new cold gas system dedicated to relative orbit control maneuvers. This choice would guarantee the availability of appropriate thrusters for relative orbit control, and a quasi-identical execution of orbit maintenance maneuvers by the two spacecraft.

Ground-in-the-loop versus Autonomous Orbit Control

The TSX mission design foresees a ground-in-the-loop orbit control system. The space segment, mainly consisting of the TSX satellite, provides GPS data (e.g. navigation solutions, code and carrier phase measurements) and AOCS housekeeping data to the ground segment during ground station contacts. These data are filtered and used for an orbit determination to get the best possible knowledge of the satellite status and motion. The control variables and the orbital element deviations from the target trajectory are calculated and handed over to the controller software. Inputs for the orbit determination and controller software are also external data, consisting of up to date solar flux data, GPS auxiliary data and Earth rotation parameters from the International Earth Rotation Service (IERS). The controller compares the predicted orbit parameters over the next 24 h with the control deadband and calculates a time-tagged maneuver, which is uploaded in the next ground station contact. The TSX flight dynamics system is based on a Linux PC cluster architecture with all software running autonomously, controlled by bash and Perl scripting. There is no manual interaction for the TSX orbit control during the nominal mission phase.

The ground station contacts are limited due to geographic position of the station and the costs for contact time. Only with a polar ground station a contact visibility is possible every orbit for LEO satellites. TSX uses only the Weilheim ground station (in the southern part of Germany) during routine operations. This station allows two scheduled contact per day for the nominal orbit configuration, meaning that the satellite conditions can be checked with an interval of 12 hours. While this limitation is usually not critical for single satellite operations, the visibility constraints drive the achievable orbit control accuracy for a LEO formation if a ground based approach is chosen. First of all the maneuver commands can only be uploaded twice a day. The orbit prediction arc is 12 hours long, and induces an important along-track position uncertainty[13]. Secondly the maneuver execution errors affect the relative motion, in particular the along-track separation, and can be compensated only after a ground station contact. The expected maneuver delta-v range for the TDX/TSX formation flying mission is 0.5 – 5 cm/s[11]. The lowest thrust level is required by the fine relative orbit control maneuvers, while the largest thrust level is required for orbit keeping corrections during high solar activity (end of TSX mission lifetime). Considering the TDX/TSX formation altitude, a permissible maneuver execution error of 0.25mm/s in flight direction results in a change of the semi-major axis of roughly 0.5m, and produces an along track drift of 5m per revolution (or 75m/day). This would results in a severe limitation of the along-track control accuracy for a ground-in-the-loop system. The maneuver execution error for a typical TDX maneuver pair would amount to roughly 0.35mm/s, giving an along-track control accuracy larger than 200m with a 2 days maneuver cycle. This restriction could be overcome by a fully autonomous approach. Assuming the same actuators performance (< 0.25mm/s), and a maneuver cycle of 3 orbital revolutions, an autonomous orbit control system would lead to an along-track control accuracy ten times smaller (i.e. around 20m).

Nevertheless the complex on-board processing and the lack of operational experience with autonomous close formations suggest an hybrid system and a stepwise acquisition of the nominal TDX/TSX configuration.

Flight Dynamics Operations and Contingency Scenarios

The control concept for the TDX/TSX formation is under evaluation within the TanDEM-X Phase-A study. The final solution will be the best tradeoff between a ground based and an autonomous concept. The TSX flight dynamics system could be duplicated and adapted to satisfy the formation requirements. After a housekeeping data dump, an orbit determination using GPS data is performed for both satellites. Using this orbit information, the ground-in-the-loop orbit control generates maneuver commands based on the planning cycle until the next necessary maneuver and considering the available ground station contacts. These maneuver commands are uploaded to both satellites in the same ground station contact, assuming two antennas for uplink.

The along-track separation control could be autonomously performed on-board the TDX satellite to cope with the slow ground reaction time. If short along-track baselines (e.g. few tens of meters) between the satellites are required for SAR interferometry purposes, an autonomous control becomes essential, being the longitude separation the most sensitive control variable. Special emphasis has to be given to the contingency scenarios. Keeping in mind that TSX is passive from a formation control point of view, we can individuate 4 contingency cases:

a) TSX does not execute a commanded orbit keeping maneuver,

b) TDX does not execute a commanded orbit keeping maneuver,

c) Both satellites do not execute a commanded orbit keeping maneuver,

d) TDX does not execute a commanded relative orbit control maneuver.

In case a) and b) the e-/i-vector separation is affected by the non-zero semi-major axis difference that is temporarily introduced by unsynchronized maneuvers. The extreme case of a 5 cm/s orbit keeping maneuver introduces a 100m semi-major axis offset. Assuming an eccentricity offset of 300m, the minimum possible separation would thus be reduced to 200m. Accordingly a slightly larger eccentricity offset may be advisable near the mission end. Although the motion in the plane perpendicular to the flight direction is safe, the along-track separation drifts severely and could cause the violation of the formation configuration requirements. TSX in case a), or TDX in case b), have to execute the missed maneuver. Furthermore TDX has to perform corrective maneuvers to restore the nominal separation. The active relative motion correction should be autonomous, to avoid a late ground response.

In case c) the formation relative configuration results unperturbed. The TSX distance from the target trajectory would exceed the limits, but the next ground contact can be used for corrections. In case d) there are no issues in the plane perpendicular to the flight direction because of the collision avoidance strategy. The longitude control window could be violated if the ground reaction is slow and there is not autonomous reaction.

CONCLUSION

The e-/i-vector separation method is shown to be a powerful tool for formation flying design and proximity analyses. The presented investigation demonstrates that a description of the linearized relative motion equations in terms of Keplerian elements differences is suitable either for synthesizing passive apertures or for designing active closed loop relative position control. The study presents a comprehensive verification of the proposed formation flying concept. The results focus on a numerical validation of the relative orbit control. Nonlinear simulations making use of state-of-the-art dynamic models confirm the analytical assessment and show that simple techniques can meet the demanding requirements of the long-term close formation flying. The operational implications have been briefly discussed. Although an on-board autonomous orbit control could decrease the mission cost and allow improved baselines, it represents a new system architecture that poses challenges in the areas of onboard sensing and actuation. On the other hand a conservative ground-in-the-loop system could not be feasible, due to the lack of continued ground station coverage and orbit prediction accuracy. The best tradeoff is nowadays a stepwise approach to the close formation flying where ground-based and on-board tasks are harmonized to the largest extent.

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Formation Flying Concept for Close Remote Sensing Satellites

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